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A PRELIMINARY ANALYSIS OF ORBIT INSERTION GUIDANCE FOR THE VOYAGER MISSION

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ABSTRACT

The terminal orbit insertion guidance maneuver for the Voyager mission was investigated. The study consists of two parts, the first a comparison of four types of possible guidance laws in terms of propellant efficiency with an optimum thrust program, the second an analysis of the accuracy characteristics of two of these schemes. This analysis was done by introducing navigation and implementation errors into these two guidance laws and observing the resulting error in the parameters of the terminal orbit. The gravity-turn law and constant direction in inertial space law, which compared favorably with the optimum thrust program, were chosen from part one to be used in part two. Both laws responded adequately to the induced errors without critical results. It was concluded that the preferred law would probably be the constant inertial law because it is the easier to mechanize.

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SUMMARY

The terminal orbit insertion guidance maneuver for the Voyager mission was investigated. The study consists of two parts, the first a comparison of four types of possible guidance laws in terms of propellant efficiency with an optimum thrust program, the second an analysis of the accuracy characteristics of two of these schemes. This analysis was done by introducing navigation and implementation errors into these two guidance laws and observing the resulting error in the parameters of the terminal orbit. The gravity-turn law and constant direction in inertial space law, which compared favorably with the optimum thrust program, were chosen from part one to be used in part two. Both laws responded adequately to the induced errors without critical results. It was concluded that the preferred law would probably be the constant inertial law because it is the easier to mechanize.

I. INTRODUCTION

With the rapid advent of more ambitious interplanetary space missions, the guidance mode required for the transfer from an approach hyperbola to an elliptical orbit about a planet needs additional consideration. This study, which concerns such an investigation, also serves in part to verify the results of previous work done in this area.

The first part of this study is concerned with the question of what guidance laws are feasible in terms of propellant requirements for insertion into a terminal orbit about Mars. In an attempt to answer this question, a two-dimensional analysis was made by comparing the payload and ΔV losses for each of four types of guidance laws with an optimum thrust program. The second part is an error analysis and an accuracy study for two of these laws applied to orbit insertion guidance. Each of these two laws was simulated in a two-dimensional analysis with induced navigation and implementation errors, and the resulting errors in the parameters of the terminal ellipse were observed. A gravity-turn law and a constant direction fixed in inertial space law were selected from the first part to be used as the guidance laws in the error analysis.

The results of the first part of the study show that the gravity-turn law gives the minimum losses with respect to the optimum thrust program. However, because of the short-burn arc, all three non-optimum guidance laws compare very favorably with the optimum thrust program. The error analysis shows that both laws have particular errors which they tolerate better than the other so that one may be considered just as accurate as the other. The problem then is reduced to a matter of judgement, the deciding factor being the ease with which the guidance law can be mechanized.

II. GENERAL DISCUSSION

This report seeks to investigate, not in a highly sophisticated study but rather in a simple yet informative type of study, two principal questions:

(1) Can a guidance law be found which is easy to mechanize and still be feasible in terms of propellant used for insertion into a terminal orbit?

(2) If found, can such a law provide the necessary accuracy for the mission?

The methods of approach and the answers to these two questions are given in the following discussions.

A. COMPARISON OF GUIDANCE LAWS

For this phase of the study, the four thrust-vector pointing laws used are described as follows:

(1) The calculus of variations optimum thrust program

(2) Gravity-turn (the thrust-vector reverse and collinear to the velocity vector)

(3) Constant direction in inertial space

(4) Constant local pitch (the thrust-vector perpendicular to the instantaneous radius vector).

For the entire study a nominal elliptical orbit of 1,000 by 10,000 kilometers was chosen. Other pertinent data for the study may be found in Table 1. The nominal point of insertion into the elliptical orbit was chosen to be periapsis, and the C_3 for the nominal hyperbola was chosen to be 10.356184 km^2/sec^2.

Trajectories for each guidance law were generated by starting at periapsis on the ellipse and integrating backward until the C_3 (twice the energy per unit mass) corresponding to the approach hyperbola was reached.

The minimum velocity transfer for each guidance law was found by biasing each law by plus and minus a few degrees until a minimum time was found. The ΔV losses of the guidance law are defined to be the minimum velocity transfer minus the minimum velocity required for the optimum thrust program (see Table 2).

B. ERROR ANALYSIS

The propellant efficiency study shows that the gravity-turn law and the constant inertial law compared most favorably with the optimum program. These two laws then were chosen for investigation to determine how accurately the desired orbit is achieved when errors are induced into the system. The four sources of errors considered are listed as follows:

(1) Errors in the magnitude of the impact parameter (B) (see Figure 1).

(2) Errors in the time of encounter with the planet (in this case Mars).

(3) Errors in the parallel component of the specified velocity increment.

(4) Errors in the normal component of the specified velocity increment.

The parameters investigated with the simulation of these errors are the changes in the radius of periapsis and apoapsis and the argument of periapsis (ω) of the nominal ellipse.

To simulate the error in the magnitude of the impact parameter (B), it was necessary to derive a relationship between B and the radius vector to the nominal point for starting the retro-burn into the elliptical orbit. In essence, then, an error in the magnitude of B results in an error at arrival at the correct position of the nominal starting point for the retro-burn. The results of these errors are shown in Figures 2 and 2A.

In simulating the errors in the time of encounter (the time at which the spacecraft arrives at periapsis of the approach hyperbola without terminal thrusting), it was assumed that a time error between the nominal starting point for retro-burn and the time of encounter maps over 1 to 1. Therefore, to have a resulting error of 30 seconds in the time of encounter, it would be necessary to vary by 30 seconds the time to begin the retro-burn. The resulting changes in the nominal elliptical orbit caused by errors in the time of encounter are shown in Figures 3 and 3A.

To induce a velocity error in the parallel component of the specified velocity increment, it was necessary to start at the nominal starting point for the retro-burn and vary the thrust magnitude by certain percentages. Then, at cutoff time (insertion point) the resulting velocity errors can be interpreted. Errors in the parallel component of velocity were taken to correspond with certain allowable percentages of the total specified retro-velocity increment. Figures 4 and 4A show the changes in the parameters of the nominal elliptical orbit versus percent changes in thrust magnitude.

Errors in the normal component of the specified velocity increment were simulated by thrust misalignment. The thrust-vector at the nominal starting point for retro-burn was misaligned in varying directions to induce some velocity in the normal component, which is zero under nominal conditions at the insertion point. The errors in the normal component were again taken to be certain allowable percentages of the total specified retro-velocity increment. These errors then are represented by errors in the pointing of the thrust-vector as shown in Figures 5 and 5A.

Tables 3 and 4, which show the results taken from the graphs, correspond, respectively, to the maximum allowable errors (navigation and implementation errors) and the design goal errors specified for the Voyager mission in Reference 1. These tables also compare the gravity-turn law and the constant inertial guidance law to determine which would give the more accurate orbit under the influence of the simulated errors. Here again periapsis, apoapsis, and argument of periapsis are the pertinent parameters that are compared.

III. CONCLUSIONS

Many observations may be made from this study. First, for the Voyager mission, it would seem that the navigation and implementation errors as specified by the range of allowable and design goal errors are within the capabilities of both the gravity-turn law and constant inertial guidance law. It was observed that periapsis never falls below 630 kilometers for any of the maximum allowable errors; thus, it is well above the minimum altitude of the 500 kilometers required for quarantine purposes.

Second, the analysis of the guidance laws indicates that an easily mechanized guidance law is sufficient to perform the insertion maneuver in a near-optimum fashion. The optimum thrust program gives the best results, but it is very difficult and costly to mechanize. The gravity-turn law gives results very close to optimum, but it is also difficult to mechanize since the direction of the instantaneous velocity vector in inertial space must always be known. The constant direction in inertial space gives the next best results (see Table 2), which are not too far from optimum. This law is the more easily mechanized system, and since it gives approximately the same accuracies in the error analysis, it would probably be preferable. The constant local pitch scheme results in the highest performance losses.

Third, the design and maximum allowable errors seem to be tolerated by the guidance laws without any critical results. Other studies have suggested that a loose insertion orbit about Mars which will be compensated for by orbit trim maneuvers will be allowed. If this be the case, then this study indicates that the magnitude of the required retro-velocity insertion increment for our nominal orbit may be sufficiently controlled by a timer to effect capture rather than by an accelerometer for terminating the braking thrust.

REFERENCES

1. JPL, SE 002 BB 001-1B21, "Performance and Design Requirements for the 1973 Voyager Mission General Specification for," Oct. 17, 1966.

2. JPL EPD-250, "Mariner Mars 1969 Orbiter Technical Feasibility Study," November 16, 1964.

TABLE 1

1.	Radius of Mars	3400 km
2.	Gravitation Constant for Mars	42883 km^3/sec^2
3.	C_3 of Approach Hyperbola	10.356184 km^2/sec^2
4.	Periapsis Altitude of Nominal Ellipse	1000 km
5.	Apoapsis Altitude of Nominal Ellipse	10,000 km
6.	Thrust	7750 lbs
7.	Specific Impulse	300 sec.
8.	Weight in Terminal Orbit	11,400 lbs
9.	Nominal Burn Time	330 sec.

TABLE 2

Guidance Law	Minimum Transfer (m/s)	Guidance Law Losses (m/s)	Mass Fraction (W_f/W_o)
Impulsive	1632.7	N/A	.57934
Optimum Thrust Program	1637.2	0	.57320
Gravity-Turn	1637.3	.1	.57318
Constant Inertial Direction	1638.6	1.3	.57296
Constant Local Pitch	1640.6	3.3	.57257

TABLE 3

ERRORS RESULTING FROM A SIMULATION OF THE MAXIMUM ALLOWABLE ERRORS FOR A CONSTANT INERTIAL DIRECTION AND GRAVITY TURN GUIDANCE LAWS

	Impact Parameter Error (B) $\Delta B = \pm 300$ km		Error in Encounter Time $\Delta t = \pm 120$ secs.		Error in Component Parallel to Specified Velocity Increment $\Delta V = \pm 3\%$*		Error in Component Normal to Specified Velocity Increment $\Delta V = \pm 5\%$*	
	Constant Inertial	Gravity Turn	Constant Inertial	Gravity Turn	Constant Inertial	Gravity Turn	Constant Inertial	Gravity
Δr_P	+360 km -360 km	+368 km -354 km	-9.0 km -5.5 km	-6.0 km -56.0 km	+2.0 km -2.0 km	+2.0 km -2.0 km	-15 km +5.0 km	-15 km +7.5 km
Δr_A	+7400 km -4100 km	+7338 km -4170 km	-165 km +649 km	+16.0 km +357.0 km	-1250 km +1500 km	-1250 km +1500 km	-100 km +300 km	+50 km +50 km
ω	$+1.5^\circ$ -2.4°	$+.90^\circ$ -2.4°	-22° -4.5°	-25.3° -8.1°	$-.2^\circ$ 0	$-.25^\circ$ $-.50^\circ$	-3.5° -3.5°	-2.5° -2.5°

* Percent of Specified Velocity Increment

TABLE 4

ERRORS RESULTING FROM A SIMULATION OF THE DESIGN GOAL ERRORS FOR A CONSTANT INERTIAL DIRECTION AND GRAVITY TURN GUIDANCE LAWS

	Error in Impact Parameter (B) $\Delta B = \pm 100$ km		Error in Time of Encounter $\Delta t = \pm 30$ sec.		Error in Component Parallel to Specified Velocity Increment $\Delta V = \pm 1.5\%$*		Error in Component Normal to Specified Velocity Increment $\Delta V = \pm 3\%$*	
	Constant Inertial	Gravity Turn	Constant Inertial	Gravity Turn	Constant Inertial	Gravity Turn	Constant Inertial	Gravity Turn
Δr_P	+125 km -125 km	+121 km -125 km	+2 km -2 km	+4 km -8 km	+1 km -1 km	-1 km +1 km	-10 km +5 km	-10 km +5 km
Δr_A	+2000 km -1600 km	+1962 km -1630 km	-50 km +100 km	-31 km +54 km	-650 km +700 km	-650 km +650 km	-50 km +150 km	+25 km +25 km
$\Delta \omega$	$+.4^\circ$ $-.8^\circ$	$+.314^\circ$ $-.786^\circ$	-7° -1.8°	-6.37° -2.6°	$\sim 0^\circ$ $\sim 0^\circ$	-1.7° -2.5°	-2.0° -2.0°	-1.5° -1.5°

* Percent of Specified Velocity Increment

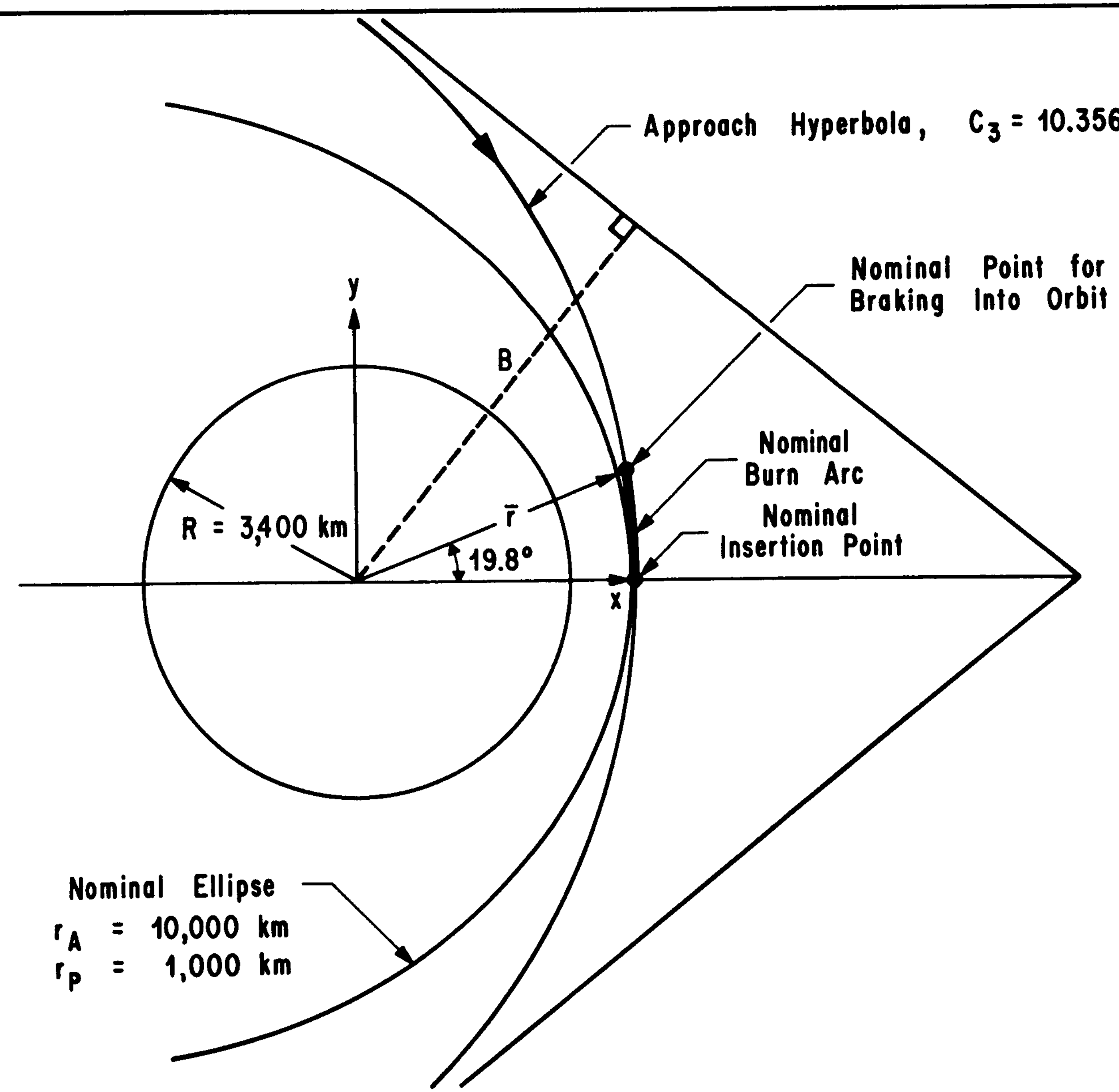

FIG. 1. TRANSFER GEOMETRY

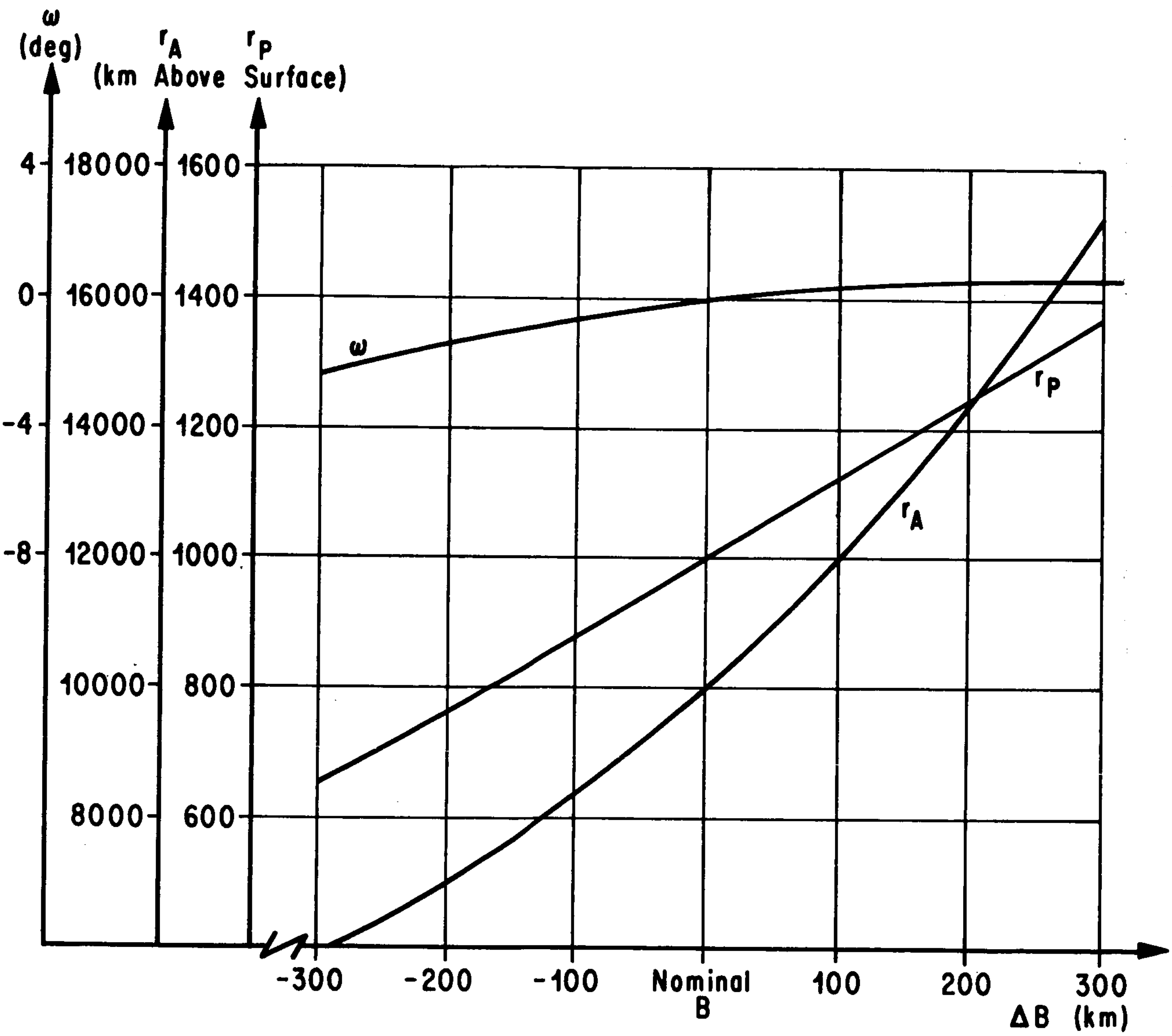

FIG. 2. ERRORS IN IMPACT PARAMETER B
(GRAVITY TURN)

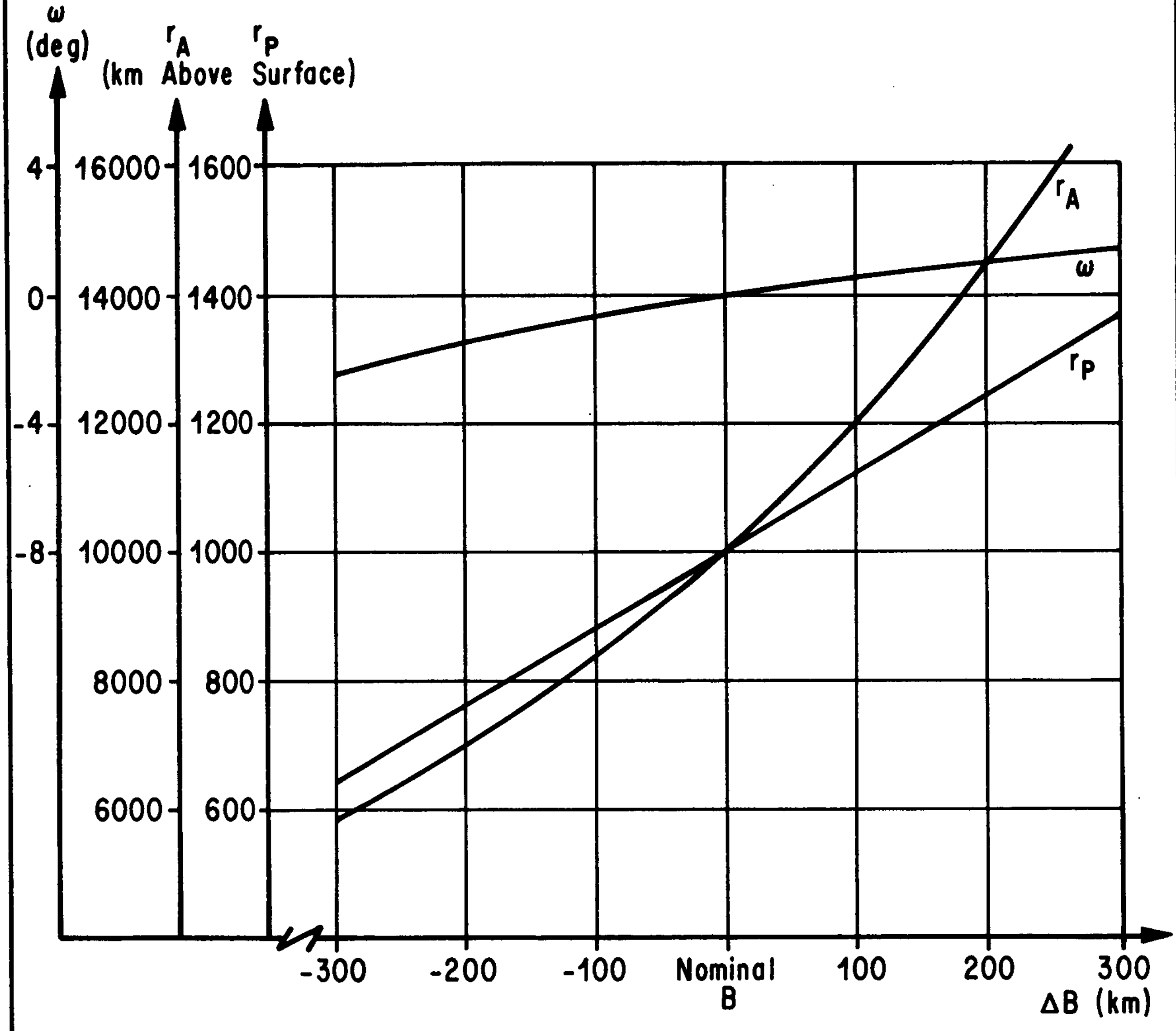

FIG. 2A. ERRORS IN IMPACT PARAMETER B
(CONSTANT INERTIAL)

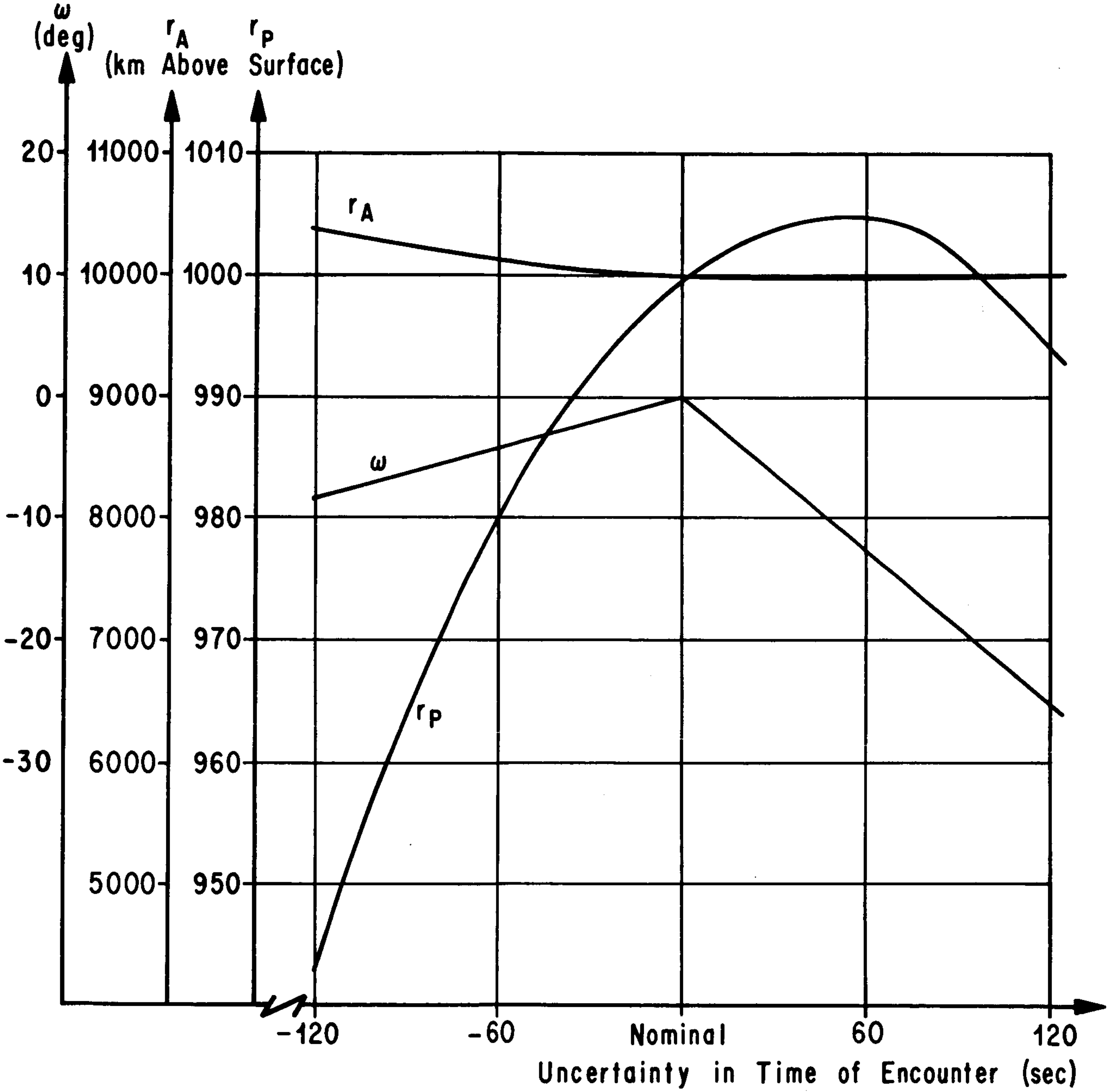

FIG. 3. ERRORS IN ENCOUNTER TIME
(GRAVITY TURN)

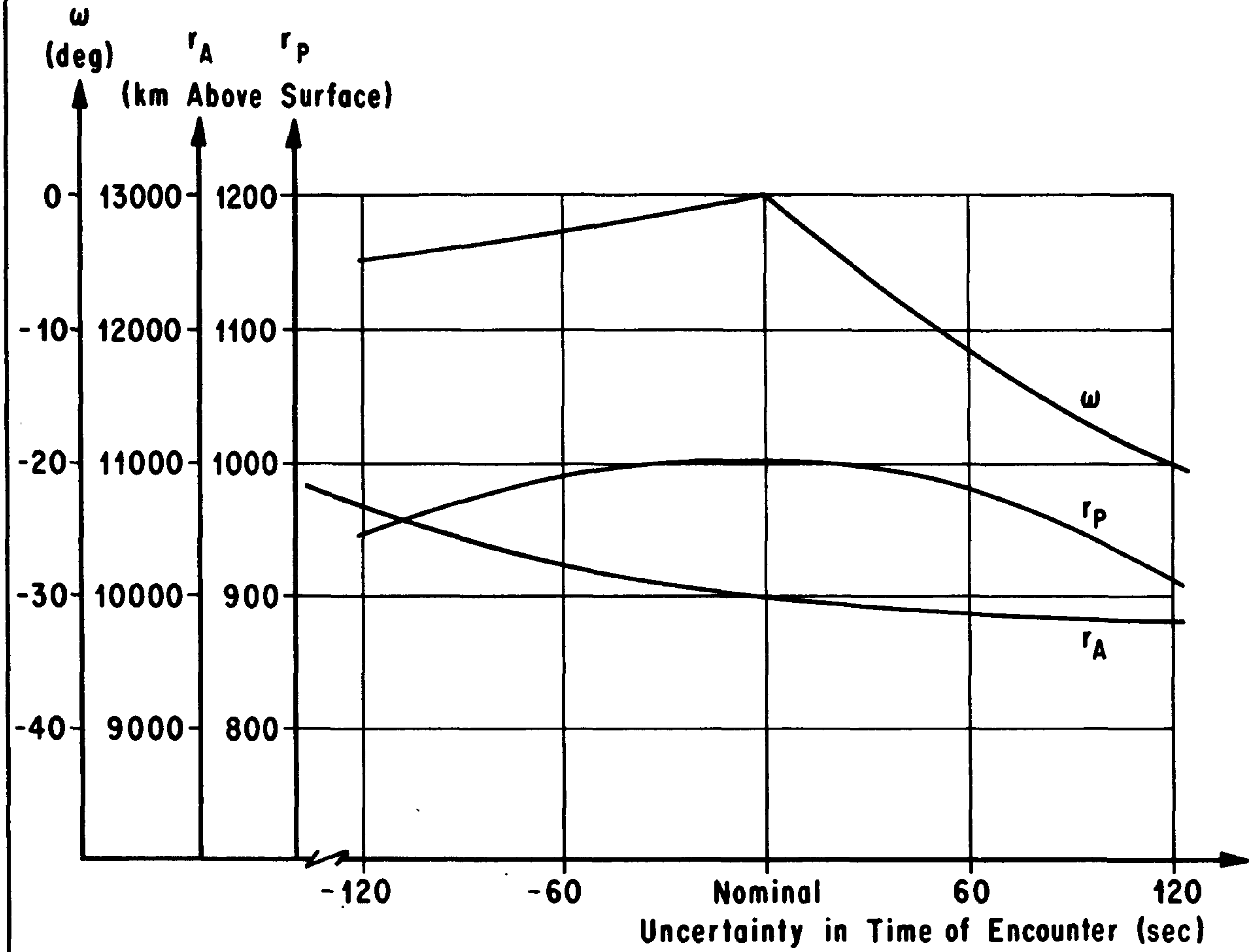

FIG. 3A. ERRORS IN TIME OF ENCOUNTER (CONSTANT INERTIAL)

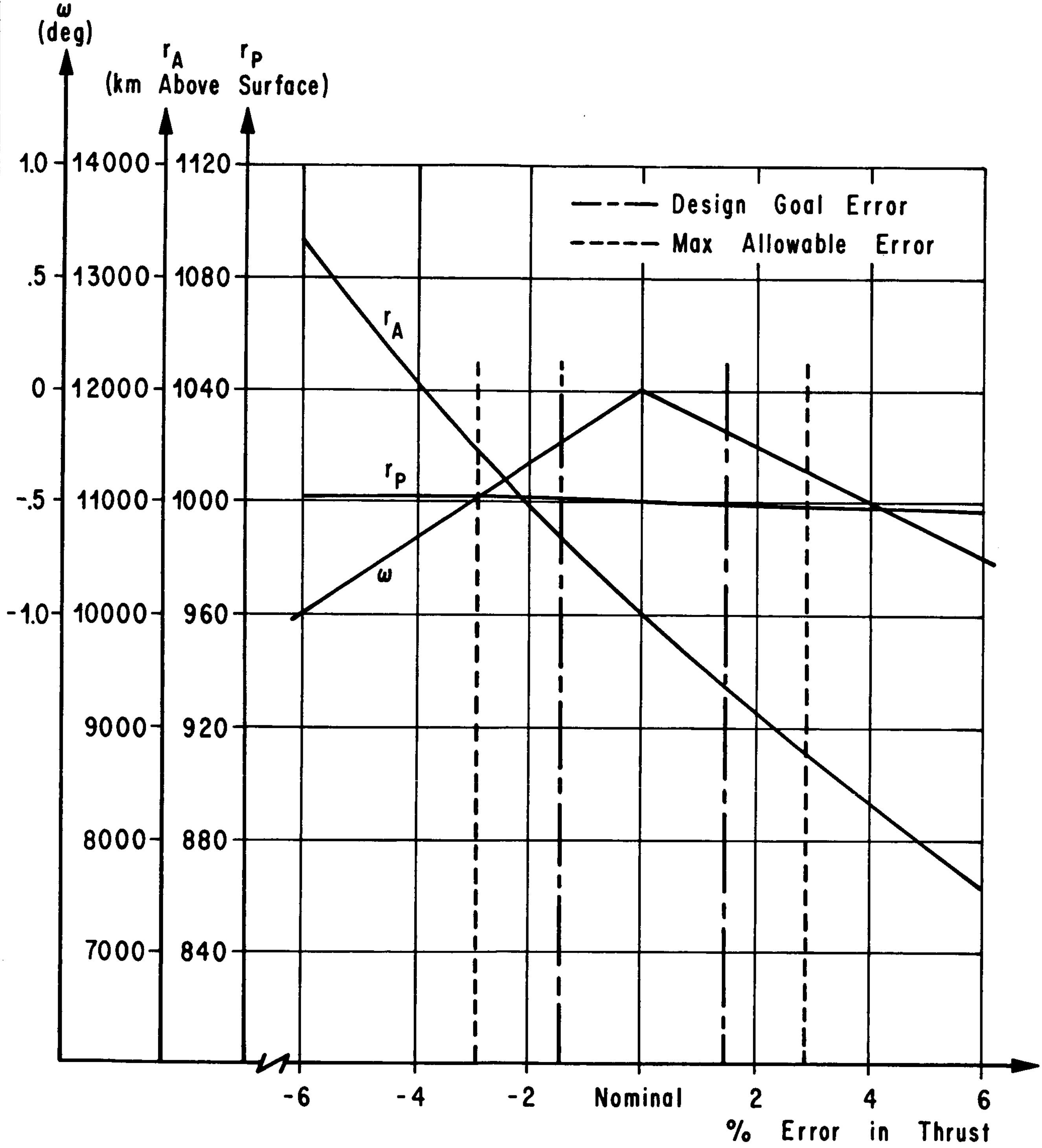

FIG. 4. ERRORS IN THRUST MAGNITUDE (GRAVITY TURN)

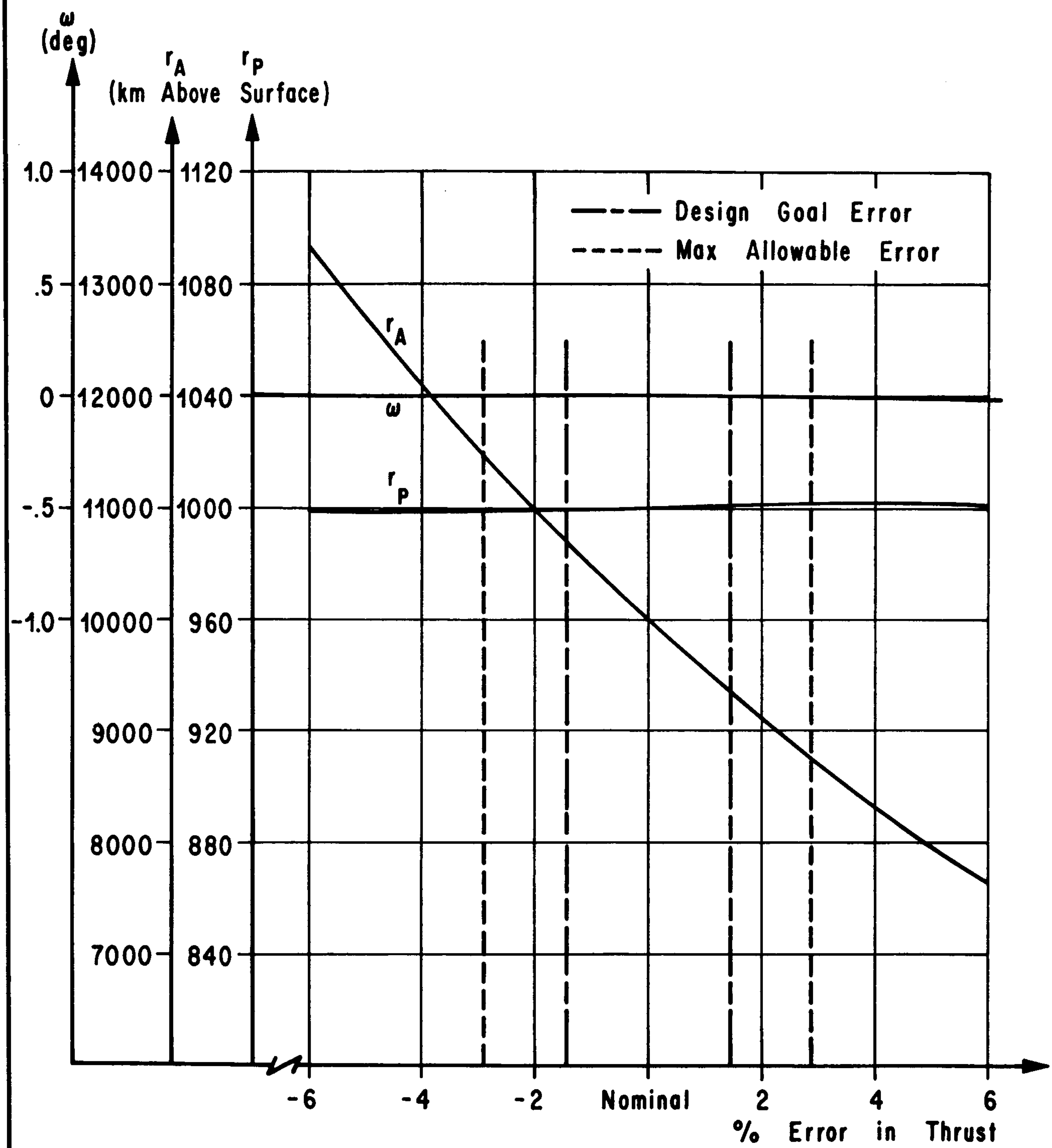

FIG. 4A. ERRORS IN THRUST MAGNITUDE (CONSTANT INERTIAL)

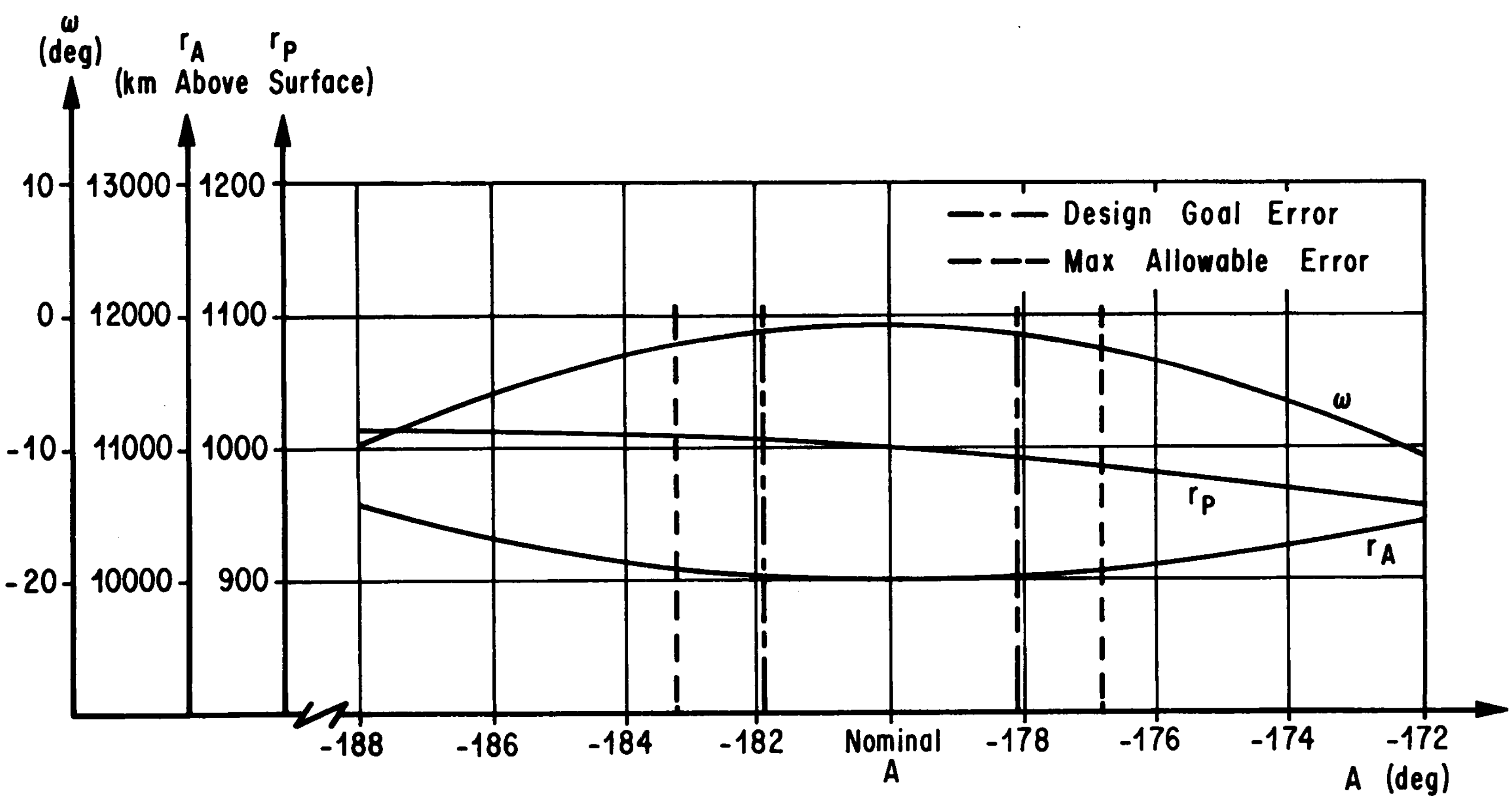

FIG. 5. ERRORS IN THRUST ALIGNMENT
(GRAVITY TURN)

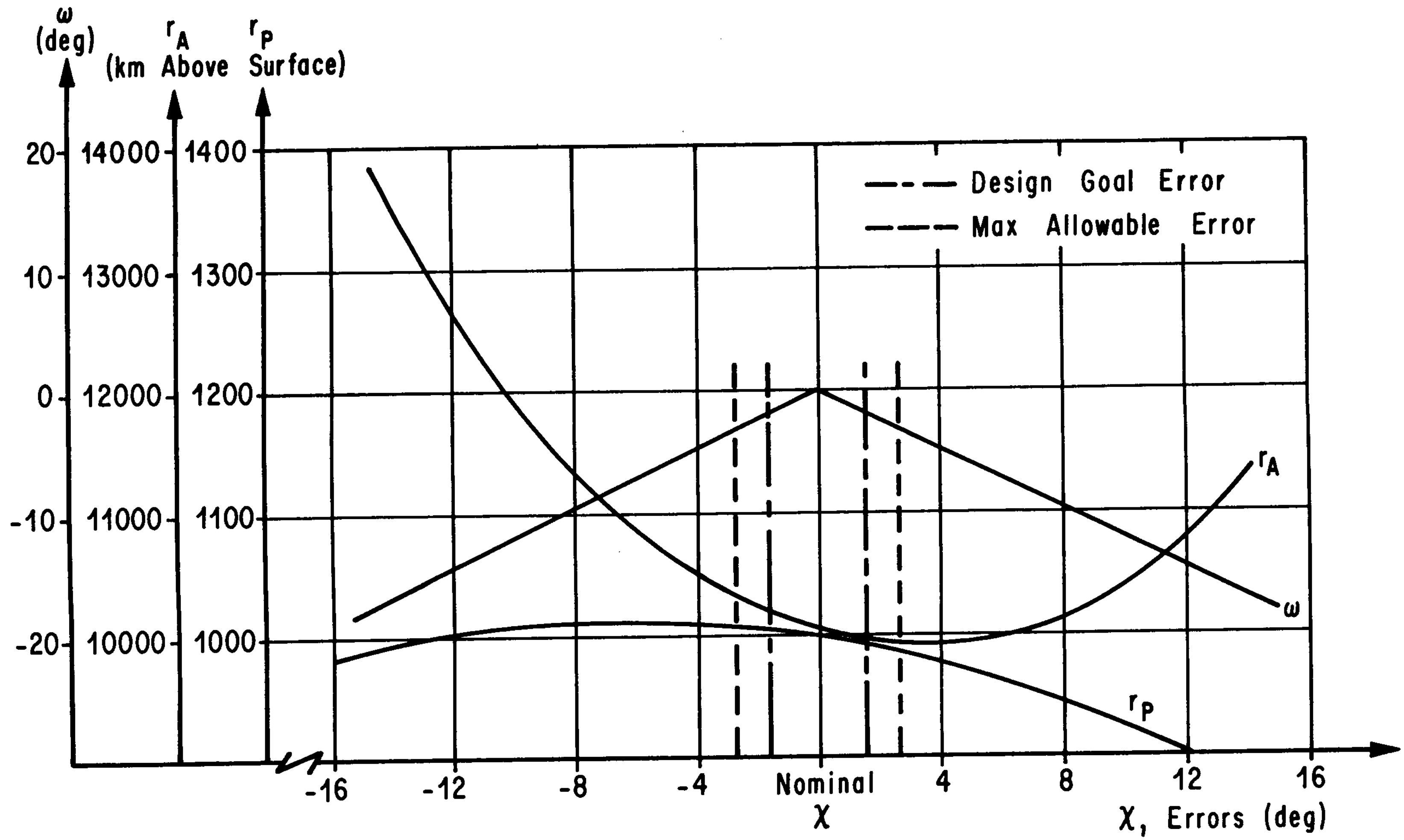

FIG. 5A. ERRORS IN THRUST ALIGNMENT
(CONSTANT INERTIAL)

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This document has also been reviewed and approved for technical accuracy.

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